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ASSESSMENT OF HIGH POWER ELECTRIC PROPULSION CONCEPTS FOR ENHANCED MISSION CAPABILITY

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SUMMARY

This report describes the development and use of the SPACEDRIVE software. SPACEDRIVE is user interactive software that determines system and mission parameters for potential SDI earth orbital electric propulsion applications. SPACEDRIVE also contains an electric propulsion reference search data base and descriptive overviews of a large number of electric propulsion engine concepts. Specific model equations contained in SPACEDRIVE are presented and their terms and use defined. Operation of each SPACEDRIVE utility is discussed.

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This report describes the development and use of the SPACEDRIVE software. SPACEDRIVE is user interactive software that determines system and mission parameters for potential SDI earth orbital electric propulsion applications. SPACEDRIVE also contains an electric propulsion reference search data base and descriptive overviews of a large number of electric propulsion engine concepts. Specific model equations contained in SPACEDRIVE are presented and their terms and use defined. Operation of each SPACEDRIVE utility is discussed.

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1.0 OBJECTIVE

Facilitate a greater general understanding of the capability of various electric propulsion concepts by developing user interactive software that determines system and mission parameters for potential SDI earth orbital electric propulsion applications.

2.0 INTRODUCTION

This document describes the functions, underlying models, and operation of the SPACEDRIVE software package which is contained in the accompanying diskettes. SPACEDRIVE utilities perform several functions. First, a series of system scaling relationships, developed primarily from performance data for existing engines, forms the basis for the SPACEDRIVE systems analysis utility for ion engines and arcjets operating in high power ranges. Second, a mission analysis utility compares application of ion and arcjet engines with chemical propulsion for several earth orbital missions. SPACEDRIVE also includes a database utility of electric propulsion technical references and provides search functions for this library, allowing the user access to performance data and projections for a wide variety of electric propulsion engine concepts. Finally, SPACEDRIVE has a descriptive overview utility which describes the principle of operation, state of development, typical performance characteristics, and high power operating potential for a wide range of electric propulsion concepts.

SPACEDRIVE makes the utilities presented above readily accessible through a structured collection of user interactive menus. These menus contain self explanatory messages describing how to use the menu and how to proceed to other menus. No prior expertise in the use of this software is required for succesful operation of SPACEDRIVE. Each SPACEDRIVE utility provides many options to the user. The versatility of SPACEDRIVE and its ability to provide propulsion system mission trade off comparisons depends on the creativity of the user in manipulating these options.

3.0 SYSTEMS ANALYSIS UTILITY

The systems analysis utility of SPACEDRIVE requires that the user input the available spacecraft power, the engine specific impulse and propellant and whether ion or arcjet engines are to be considered. These data have been chosen as SPACEDRIVE system analysis inputs because they should be readily available when the user has specific spacecraft and mission constraints with which the propulsion system must comply. When the user supplies these data, the systems analysis determines, within the limits of the model, the propulsion system design and performance specifications that satisfies the user's inputs. The resultant propulsion system performance level and mass estimates are conservative and are derived from demonstrated and well documented hardware development and flight programs.

3.1 System Analysis Assumptions

Several assumptions are included in the model development for both the ion engine and arcjet engine system analyses and are listed below:

(i) The power system is assumed to be already present as part of the host spacecraft and its mass is simply one part of the user supplied initial host spacecraft mass.

(ii) There is one power processor unit per engine. Passive thermal radiators are assumed for power processor heat rejection. The thermal radiator mass is included in the propulsion system structural mass and not in the power processor mass projection.

(iii) Power processor mass estimates include telemetry, command and control circuitry, and cable harness and connector allowances.

(iv) Attitude control is assumed to be provided by the host spacecraft attitude control system.

(v) No gimbals are used.

(vi) All engines include spacecraft thermal protection in the mass estimates.

(vii) The ion and arcjet propulsion system mass estimates do not include the propellant tank and tank supporting structure (however, the mass of these components is included in the SPACEDRIVE mission analysis utility calculations).

(viii) The flow system mass estimates include regulators, valves, filters, flow lines, etc., that are appropriate to the propellant and engine choice, but do not include the propellant tank and tank supporting structure.

(ix) No contingency allowance is provided in the ion and arcjet propulsion system mass estimates (or in the chemical propulsion system mass estimates which are used in the SPACEDRIVE mission analysis utility).

(x) Demonstrated and conservative component technology levels are assumed.

(xi) For a given available spacecraft power, the propulsion systems are sized to use the largest possible ion or arcjet engines within the constraints imposed by the model. This approach minimizes the number of ion or arcjet engines required to process a given host spacecraft available power level and also does not require the user to know, apriori, specific engine sizes and operating points.

If the user wishes to determine system level parameters for a specific ion or arcjet and then apply these to a mission, the user can enter in the system analysis the known engine design specific impulse, propellant type, and, in place of the available spacecraft power, enter a value equal to 1.11 times the engine input power (this accounts for the power processor inefficiency). The SPACEDRIVE system analysis utility will then determine all the mass and performance estimates for the specific single engine design. Appropriate multiples of this single engine system can then be specified by the user for the available spacecraft power in order to specify the mass of the user's specific multiple engine system for input into, and proper functioning of, the SPACEDRIVE mission analysis utility.

Figure 1 shows schematically the major electric propulsion system elements assumed in the SPACEDRIVE systems analysis utility. As mentioned previously, the propulsion system mass determined by the system analysis utility does not include the propellant, and propellant tank and supporting structure. This subsystem element, along with the user host spacecraft adapter mass (which is 10% of the mass of everything to its right in Figure 1) is included in the SPACEDRIVE mission analysis utility calculations.

3.2 Ion Engine Analysis Output Parameters

The ion engine system analysis calculates both engine and system design and performance parameters. A complete list of the calculated parameters follows (note that in this list each parameter has listed its units as used by the system model equations in section 3.3.2 and, in parentheses, its common usage units which are displayed on the SPACEDRIVE system analysis utility screen):

Engine Parameters:

I_{sp}	engine specific impulse, s (s)
P_E	engine input power, kW (kW)
η_E	total engine efficiency
T_E	engine thrust, N (N)
T_E/P_E	thrust-to-power ratio, N/kW (mN/kW)
J_b	beam current, A (A)

V_b	beam voltage, V (V)
V_t	total voltage, V (V)
R	net-to-total voltage ratio
J_d	discharge current, A (A)
D_b	beam diameter, m (cm)
ℓ_g	grid separation, m (mm)
d_s	screen hole diameter, m (mm)
$J_{b/H}$	beam current per hole, A (mA)
m_E	engine mass, kg (kg)
m_{pp}	power processor mass, kg (kg)

<u>System Parameters</u>:

T	total system thrust, N (N)
P	available spacecraft power, kW (kW)
T/P	thrust-to-power ratio, N/kW (mN/kW)
m_{ps}	propulsion system mass, kg (kg)
α_{ps}	propulsion system specific mass, kg/kW (kg/kW)
N_E	number of ion engines required
A_E	engine system area, m^2 (m^2)
A_{pp}	power processor system area, m^2 (m^2)
m_{fs}	flow system mass, kg (kg)
m_s	ep system structural mass, kg (kg)

$\dot{m}$ propulsion system mass flow, kg/s (g/s)

η_{ps} propulsion system efficiency

Figure 2 shows an electron-bombardment ion engine (which is the ion engine concept assumed for this analysis) and identifies the major engine design parameters that are included in the list above.

The equations used by SPACEDRIVE to define these ion engine parameters are derived primarily from physical principles and referenced performance data of existing ion engines. The equations and assumptions used in the SPACEDRIVE ion engine systems analysis are presented and discussed in the sections immediately following.

3.3 Ion Engine Analysis Model

The SPACEDRIVE ion engine system analysis is designed to give satisfactory results with a minimum of user inputs and with a minimum required level of user electric propulsion technology expertise. For users familiar with ion engine operation, operating data from existing engines may be used as inputs into the SPACEDRIVE ion engine system analysis and the results may be compared with known parameters.

3.3.1 Ion Engine System Analysis Constants

Certain physical constants and some fixed ion propulsion system parameters are used in the SPACEDRIVE ion engine system analysis utility. The nomenclature for these constants are shown below:

ϵ discharge loss (eV/ion)

V_{nc} neutralizer coupling voltage (V)

V_d discharge voltage (V)

e fundamental electric charge (c)

g_o Earth gravitational acceleration (m/s^2)

η_u total engine propellant utilization

∂ total engine thrust loss factor

E grid gap electric field stress (V/m)

m_i ion mass (kg)

η_{pp} power processor efficiency

S/G grid span-to-gap ratio

m_{ts} propellant tank and structure mass fraction

m_{cps} chemical propulsion system mass (kg)

The values assigned these constants are shown below:

$\epsilon = 150$ eV/ion (discharge loss)

$V_{nc} = 10$ V (neutralizer coupling voltage)

$V_d = 28$ V (discharge voltage)

$e = 1.6 \times 10^{-19}$ C (fundamental electric charge)

$g_o = 9.8$ m/s^2 (gravitational constant)

$\eta_u|_{X_e} = 0.90$ (xenon propellant utilization efficiency)

$\eta_u|_{K_r} = 0.88$ (krypton propellant utilization efficiency)

$\eta_u|_{A_r} = 0.78$ (argon propellant utilization efficiency)

$\partial = 0.95$ (total thrust loss factor)

$m_i|_{X_e} = 2.193 \times 10^{-25}$ kg (xenon ion mass)

$m_i|_{K_r} = 1.399 \times 10^{-25}$ kg (krypton ion mass)

$m_i|_{A_r} = 6.671 \times 10^{-26}$ kg (argon ion mass)

$\eta_{pp} = 0.90$ (power processor efficiency)

The following constraints are applied to the permissible specific impulse values:

for X_e: $2500 \leq I_{sp} \leq 10000$

for K_r: $3000 \leq I_{sp} \leq 10000$

for A_r: $3500 \leq I_{sp} \leq 10000$

where I_{sp} is the specific impulse in seconds.

3.3.2 Ion Engine Equations

This section presents the specific equations and methodology used by SPACEDRIVE to determine ion engine and system performance parameters. In the most part, these equations are presented in the order in which they are used by SPACEDRIVE. While some of the equations are derivable from known ion engine physical operating principles, many are empirically derived from scaling known ion engine performance characteristics and system characteristics. The papers and reports from which these data were obtained are contained in the reference section at the end of this document. Some assumptions had to be made to construct a closed form solution scheme. These assumptions were, in all cases, made to give conservative results with no stringent demand on any one particular technology level. It is impossible, in a general model such as this one, to embrace everyone's views on specific levels of technology development. Nevertheless, the model equations do satisfactorily predict the performance of most electron-bombardment ion engines that have been developed or are presently under development and testing. The solution scheme is as follows:

From the user inputs of available spacecraft power and desired specific impulse, an equivalent propulsion system total discharge current is calculated from:

$$J_d^* = \frac{P\,\eta_{pp}\left[\epsilon + V_d\right]}{V_d\left\{\dfrac{m_i}{2e}\left[\dfrac{g_o I_{sp}}{\eta_u \partial}\right]^2 + \epsilon + V_{nc}\right\}}\ , \text{A}$$

where J_d^* is the equivalent total discharge current for the entire propulsion system.

Next, the number of ion engines necessary to process this available spacecraft power at the given specific impulse is determined by assuming a maximum discharge current per engine of 100 ampere and substituting into:

$$N_E = \frac{J_d^*}{J_{d\,max}}$$

where N_E is the number of engines, which is set to the next highest integer and $J_{d\,max}$ is the maximum engine discharge current (A).

Note that no engine redundancy is assumed and that for the rest of the analysis an equal number of ion engines and power processor units is assumed.

The input power required by each engine is determined from the power processor efficiency and the equation:

$$P_E = \frac{\eta_{pp} P}{N_E}, \text{ kW}$$

where P_E is the input power to each engine.

The engine discharge current is calculated from the equation:

$$J_d = \frac{J_d^*}{N_E}, \text{ A}$$

where J_d is the discharge current for each engine.

The ion energy, or beam voltage, is calculated from the equation:

$$V_b = \frac{m_i}{2e}\left[\frac{g_o I_{sp}}{\eta_u \partial}\right]^2, \text{ V}$$

where V_b is the engine beam voltage.

The total ion engine efficiency is calculated from the equation:

$$\eta_E = \frac{\partial^2 \eta_u}{1 + \frac{\left[\epsilon + V_{nc} \right]}{V_b}}$$

where η_E is the total ion engine efficiency.

The ion beam current for each engine is calculated from the equation:

$$J_b = \frac{2e\eta_E \eta_u P_E \times 1000}{m_i \left[g_o I_{sp} \right]^2} \text{, A}$$

where J_b is the engine beam current.

The thrust for each engine is calculated from the equation:

$$T_E = \frac{J_b m_i g_o I_{sp}}{e \, \eta_u} \text{, N}$$

where T_E is the engine thrust.

The total propulsion system thrust is calculated from the equation:

$$T = N_E \, T_E \text{, N}$$

where T is the total propulsion system thrust.

The ion engine thrust-to-power ratio is calculated from the equation:

$$T_E/P_E = \frac{J_b \, m_i \, g_o \, I_{sp}}{e \, \eta_u \, P_E} \text{, N/kW}$$

where T_E/P_E is the ion engine thrust-to-power ratio.

The ion engine system thrust-to-power ratio is calculated from the equation:

$$T/P = \frac{N_E T_E}{P}, \text{ N/kW}$$

where T/P is the ion engine system thrust-to-power ratio.

The above equations define the operating parameters of the required ion engine. To determine the design parameters for the ion engine, some assumptions are made. First, it is assumed that the ion engine will operate in a regime where the net-to-total voltage ratio, R, will be between 0.20 and 0.90. For values of R greater than 0.55, a two-grid ion accelerator system is assumed. For values of R equal to or less than 0.55, the analysis assumes that the ion engine will have a three-grid accelerator system, and that this additional grid will increase the engine mass by 10%. Also, it is assumed that the grid gap electric field stress is constant as the grid gap increases and the screen hole diameter increases in a fixed ratio to accomodate different ion engine specific impulse inputs. Finally, due to fabrication constraints, it is assumed that the screen and accelerator grid span-to-gap ratio can be no greater that 550. Application of these assumptions is achieved by an iterative solution to the following equations, where on each iteration the value of R is decreased in steps of 0.01 from an initial value of 0.90 until all equations and constraints are satisfied:

i) $$\frac{J_b}{J_b/H} = \frac{0.67\ D_b^2}{d_s^2}$$

ii) $$\frac{J_b/H}{V_t^{1.7}} = 1.032 \times 10^{-9}$$

iii) $$\frac{\ell_g}{d_s} = 0.30$$

iv) $$\frac{V_t}{\ell_g} = 2.66 \times 10^6, \text{ V/m}$$

v) $$V_t = \frac{V_b}{R}$$

vi) $d_s \geqq 1.9 \times 10^{-3}$, m

vii) $0.2 \leqq R \leqq 0.9$

viii) for beam diameter $D_b \leqq 0.30$ m :

$$S/G \leqq \exp\left[4.793 + 5.058\, D_b \right]$$

for beam diameter $D_b > 0.30$ m :

$$S/G \leqq 550$$

where S/G is the grid set span-to-gap ratio

ix) $D_b \geqq 0.10$, m

Using the values for grid diameter, power and the number of engines determined by the scheme above, the following scaling relations are used to determine the remaining parameters:

$$m_E = 17.307 + 7.082 \, \mathrm{Ln}\left[D_b \right], \text{ kg} \qquad 0.55 < R \leqq 0.9$$

$$m_E = 1.1 \left[17.307 + 7.082 \, \mathrm{Ln}\left[D_b \right] \right], \text{ kg} \qquad 0.2 \leqq R \leqq 0.55$$

where m_E is the ion engine mass including thermal isolation.

$$m_{pp} = 0.397 \left[P_E \times 1000 \right]^{0.544}, \text{ kg}$$

where m_{pp} is the power processor mass.

$$A_E = N_E\left[2.65\, D_b^2 \right], \text{ m}^2$$

where A_e is the ion engine array area.

$$A_{pp} = N_E \left[0.2277 \; P_E \right], \; m^2$$

where A_{pp} is the power processor array area.

$$m_{fs} = 4.70 + 2.62 \; N_E, \; kg$$

where m_{fs} is the flow system mass.

$$m_s = 0.30 \left[N_E \; m_E \right] + 2.277 \left[N_E \; P_E \right], \; kg$$

where m_s is the ion engine system structural and passive thermal radiator mass.

$$m_{ps} = N_E \left[m_{pp} + m_E \right] + m_s + m_{fs}, \; kg$$

where m_{ps} is the ion propulsion system mass.

$$\alpha_{ps} = \frac{m_{ps}}{P}, \; kg/kW$$

where α_{ps} is the ion propulsion system specific mass.

$$\dot{m} = \frac{2 \; \eta_E N_E \; P_E \times 1000}{\left[g_o \; I_{sp} \right]^2}, \; kg/s$$

where $\dot{m}$ is the propulsion system mass flow rate.

$$\eta_{ps} = \eta_{pp} \; \eta_E$$

where η_{ps} is the propulsion system efficiency.

3.4 Arcjet Analysis Output Parameters

The arcjet systems analysis calculates both engine and system performance parameters. A complete list of the calculated parameters follows (note that in this list each parameter has listed its units as used by the system model equations in section 3.5.2 and in parentheses its common usage units which are displayed on the SPACEDRIVE system analysis utility screen):

Engine Parameters:

I_{sp}	engine specific impulse, s (s)
P_E	engine input power, kW (kW)
η_E	total engine efficiency
T_E	engine thrust, N (N)
T_E/P_E	thrust-to-power ratio, N/kW (mN/kW)
m_E	arcjet mass, kg (kg)
m_{pp}	power processor mass, kg (kg)

System Parameters:

T	total system thrust, N (N)
P	available spacecraft power, kW (kW)
T/P	thrust-to-power ratio, N/kW (mN/kW)
m_{ps}	propulsion system mass, kg (kg)
α_{ps}	propulsion system specific mass, kg/kW (kg/kW)
N_E	number of arcjets required
m_{fs}	flow system mass, kg (kg)
m_s	ep system structural mass, kg (kg)
$\dot{m}$	system mass flow rate, kg/s (g/s)
η_{ps}	propulsion system efficiency

Figure 3 shows a d.c. arcjet engine (which is the arcjet concept assumed for this analysis) and identifies the major engine design parameters that are included in the list above.

3.5 Arcjet Engine Analysis Model

The SPACEDRIVE arcjet engine analysis is designed to give satisfactory results with a minimum of user inputs and with a minimum required level of user electric propulsion expertise. For users familiar with arcjet engine operation, operating data from existing engines may be used as inputs into the SPACEDRIVE arcjet engine system analysis and the results may be compared with known parameters.

3.5.1 Arcjet System Analysis Constants

Certain physical constants are used in the SPACEDRIVE arcjet system analysis utility. The values assigned to these constants are shown below:

$\eta_{pp} = 0.90$ (power processor efficiency)

$g_o = 9.8\ m/s^2$ (earth gravitational acceleration)

The following constraints are applied to the permissible specific impulse values:

for NH_3: $500.0 \leq I_{sp} \leq 1100.0$, $I_{spc} = 165$

for N_2H_4: $500.0 \leq I_{sp} \leq 1050.0$, $I_{spc} = 152$

for H_2: $500.0 \leq I_{sp} \leq 2250.0$, $I_{spc} = 267$

where I_{sp} is the specific impulse in seconds and I_{spc} is the cold gas flow specific impulse in seconds.

For each propellant type, a maximum engine power is assigned, in accord with conservative current technology limits:

$$P_E^*\big|_{NH_3} = 30\ kW$$

$$P_E^*\big|_{N_2H_4} = 30\ kW$$

$$P_E^*|_{H_2} = 100 \text{ kW}$$

where P_E^* is the maximum allowable engine power.

3.5.2 Arcjet Engine Equations

This section presents the specific equations and methodology used by SPACEDRIVE to determine arcjet engine and system performance parameters. In the most part, these equations are presented in the order in which they are used by SPACEDRIVE. While some of the equations are derivable from known arcjet engine physical operating principles, many are empirically derived from scaling known arcjet engine performance characteristics and system characteristics. The papers and reports from which these data were obtained are contained in the reference section at the end of this document. Some assumptions had to be made to construct a closed form solution scheme. These assumptions were, in all cases, made to give conservative results with no stringent demand on any one particular technology level. It is impossible, in a general model such as this one, to embrace everyone's views on specific levels of technology development. Nevertheless, the model equations do satisfactorily predict the performance of most d.c. arcjet engines that have been developed or are presently under development and testing. The solution scheme is as follows:

From the user input of available spacecraft power, the number of arcjet engines necessary to process this power is determined from:

$$N_E = \frac{P\,\eta_{pp}}{P_E^*}$$

where N_E is the number of engines which is set to the next highest integer.

Note that no engine redundancy is assumed and that for the rest of the analysis an equal number of arcjet engines and power processor units is assumed.

Arcjet engine input power requirements are then determined from the following equation:

$$P_E = \frac{P\,\eta_{pp}}{N_E}, \text{ kW}$$

where P_E is the arcjet engine input power.

The power-to-mass flow rate ratio is determined next from the following set of emperically derived equations:

for NH_3: $$\left[\frac{P_E}{\dot{m}}\right] = 8.543 \times 10^{-2}\, I_{sp}^{2.055}\ , \text{kW/kg/s}$$

for N_2H_4: $$\left[\frac{P_E}{\dot{m}}\right] = 5.775 \times 10^{-2}\, I_{sp}^{2.151}\ , \text{kW/kg/s}$$

for H_2: $$\left[\frac{P_E}{\dot{m}}\right] = 1.230 \times 10^{-3}\, I_{sp}^{2.608}\ , \text{kW/kg/s}$$

where $\left[\frac{P_E}{\dot{m}}\right]$ is the power-to-mass flow rate ratio.

From the above results, the arcjet engine efficiency and the arcjet propulsion system efficiency are determined using the following equations:

$$\eta_E = \frac{I_{sp}^2}{\left[\frac{P_E \times 1000}{\dot{m}}\right]\left\{\frac{2}{g_o^2} + \frac{I_{spc}^2}{\left[\frac{P_E \times 1000}{\dot{m}}\right]}\right\}}$$

where η_E is the arcjet engine efficiency.

$$\eta_{ps} = \eta_{pp}\ \eta_E$$

where η_{ps} is the arcjet propulsion system efficiency.

The remaining parameters are determined from the following scaling relations:

$$m_E = 0.0635 \left[P_E \times 1000 \right]^{0.50}, \text{kg}$$

where m_E is the arcjet engine mass including thermal isolation.

$$m_{pp} = 0.115 \left[P_E \times 1000 \right]^{0.544}, \text{ kg}$$

where m_{pp} is the power processor mass.

$$m_{fs} = 5.80 + 1.54\, N_E \text{ , kg}$$

where m_{fs} is the flow system mass.

$$m_s = 0.30 \left[N_E\, m_E \right] + 2.277 \left[N_E\, P_E \right] \text{ , kg}$$

where m_s is the arcjet engine system structural and passive thermal radiator mass.

$$m_{ps} = N_E \left[m_{pp} + m_E \right] + m_s + m_{fs} \text{ , kg}$$

where m_{fs} is the propulsion system mass.

$$\alpha_{ps} = \frac{m_{ps}}{P} \text{ , kg/kW}$$

where α_{ps} is the arcjet propulsion system specific mass.

for NH_3: $$\dot{m} = \frac{N_E\, P_E}{\left[8.543 \times 10^{-2}\, I_{sp}^{2.055} \right]} \text{ , kg/s}$$

for N_2H_4: $$\dot{m} = \frac{N_E\, P_E}{\left[5.775 \times 10^{-2}\, I_{sp}^{2.151} \right]} \text{ , kg/s}$$

for H_2: $$\dot{m} = \frac{N_E\ P_E}{\left[\ 1.230 \times 10^{-3}\ I_{sp}^{2.608}\ \right]}\ ,\ \mathrm{kg/s}$$

where $\dot{m}$ is the arcjet propulsion system mass flow rate.

$$T_E = \dot{m}\ g_o\ I_{sp}\ ,\ \mathrm{N}$$

where T_E is the arcjet engine thrust.

$$T = N_E\ T_E\ ,\ \mathrm{N}$$

where T is the total propulsion system thrust.

$$T_E/P_E = \frac{\dot{m}\ g_o\ I_{sp}}{P_E}\ ,\ \mathrm{N/kW}$$

where T_E/P_E is the arcjet engine thrust-to-power ratio.

$$T/P = \frac{N_E\ T_E}{P}\ ,\ \mathrm{N/kW}$$

where T/P is the arcjet engine system thrust-to-power ratio.

3.6 Mission Analysis

Missions were selected for analysis in SPACEDRIVE by merit of their interest to SDI applications. These missions include orbit transfer, station keeping, and station keeping with defensive maneuvering. To use the SPACEDRIVE mission analysis utility, the user specifies the pertinent mission parameters such as initial and final altitudes and mission duration. The user may select one of two on-board power sources, solar arrays or nuclear power. After selecting these mission parameters, the user is required to specify several propulsion system parameters. It is advised, but not required, that the user enter propulsion system data obtained from the SPACEDRIVE ion or arcjet system analysis. After specifying all requested data, the analysis calculates the propellant consumed on the mission and the required initial spacecraft mass including the electric propulsion system mass and propellant (initial spacecraft mass, which must include the power source, is a user required input). In addition, the total impulse, total burn time, number of earth orbits or defensive maneuvers, and the total trip time are also determined. A comparison with chemical propulsion is also provided, where for orbit transfer missions it is assumed that a cryogenic oxygen/hydrogen system is used with a vacuum specific impulse of 444 sec. For station keeping missions, a storable bipropellant NTO/MMH system is assumed with a vacuum specific impulse of 308 sec.

The station keeping mission with defensive maneuvering should be of particular interest to SDI applications. This mission simulates the flight of a spacecraft in a station keeping orbit where some slight orbit raising and then lowering back to the specified orbit is required periodically to reduce the risk of detection. As modeled in SPACEDRIVE, these defensive maneuvers occur at user specified intervals and for a user specified altitude change.

3.6.1 Auxiliary Mission Equations and Assumptions

For all ion and arcjet propulsion systems used in the SPACEDRIVE mission analysis utility, the propellant tank mass and associated support structure mass are determined by the following relations:

$$X_e: \quad m_{ts} = 0.144\ m_p,\ \text{kg}$$

$$K_r: \quad m_{ts} = 0.180\ m_p,\ \text{kg}$$

$$A_r: \quad m_{ts} = 0.250\ m_p,\ \text{kg}$$

$$NH_3: \quad m_{ts} = 0.17\ m_p,\ \text{kg}$$

$$N_2H_4: \quad m_{ts} = 0.17\ m_p,\ \text{kg}$$

$$H_2: \quad m_{ts} = 0.35\ m_p,\ \text{kg}$$

where m_{ts} is the propellant tank and tank structure mass and m_p is the propellant mass.

The additional structure mass represented by these equations is added to the overall electric propulsion system mass during the mission analysis calculations. Note that the electric propulsion system mass required as an input on the SPACEDRIVE mission analysis menu does not include the propellant tank mass and support structure since the mission propellant requirement is not known apriori. The chemical propulsion system mass estimates are determined from the following relations:

$$\text{for } O_2/H_2: \qquad m_{cps} = 1998 + 1.0589\ m_p,\ \text{kg}$$

for NTO/MMH: $m_{cps} = 89 + 1.146\ m_p$, kg

where m_{cps} is the chemical propulsion system mass which includes all hardware and propellant.

For both electric and chemical propulsion mission analyses, SPACEDRIVE adds on adapter to the user host spacecraft. The mass of this propulsion system attachment adapter is 10% of the attached propulsion system (electric or chemical) including all structure and mission propellant.

Significant simplifying assumptions used in the SPACEDRIVE mission analysis utility are given below:

(i) Atmospheric drag is neglected.

(ii) Only coplanar altitude orbit changes are allowed.

(iii) For solar array power, the spacecraft is assumed to always pass through the shadow cone of the earth as it spirals out, during which time the propulsion system is shutdown.

All references used to construct the mission models are contained in the reference listing at the end of this document.

3.6.2 Orbit Transfer Analysis

The rocket equation is used to obtain an initial estimate for the amount of propellant required to complete the mission. This initial propellant estimate is used to determine an initial spacecraft mass. This spacecraft is then subjected to the SPACEDRIVE mission analysis which calculates propellant consumption and altitude change for each orbit, iterating through the following calculations:

The spacecraft's altitude determines the required orbital velocity and the orbit period. The propulsion system's mass flow rate is used to calculate the total amount of propellant that can pass through the propulsion system on each orbit. For solar powered spacecraft, this burn per orbit is adjusted to account for the time in the earth's shadow. Having determined the propellant burned per orbit, the altitude change per orbit is then determined by numerically integrating the radial rate equation that describes the effect of essentially continuous tangential thrust applied to a quasi-circular orbit. At the end of each orbit, a check on the remaining propellant is made. If the spacecraft runs out of propellant before the final altitude is reached, then more propellant is added to the initial propellant mass estimate (and the tankage mass estimate is adjusted) and this new spacecraft is subjected to the orbit-by-orbit analysis. If the spacecraft achieves its final orbit altitude before running out of propellant, then the remaining propellant is checked. If more than 5% of the initial propellant remains in the spacecraft, then the initial propellant mass is reduced (and the tankage mass estimate is adjusted) and the new spacecraft is subjected to the orbit-by-orbit mission analysis. This process continues until the initial propellant estimate agrees to within 5% of the required propellant. When this agreement is acheived, the solution is considered acceptable.

3.6.3 Station Keeping Analysis

To maintain a spacecraft in orbit at the specified altitude, it is assumed that 50m/s/year of delta-V is required to account for the average of the sum of the north-south and east-west station keeping requirements. Using this assumption and the propulsion system mass flow rate, the propellant required to maintain the orbit for the desired number of years is readily determined. The mission analysis results follow directly.

3.6.4 Station Keeping with Defensive Maneuvering Analysis

Station keeping with defensive maneuvering combines the orbit raising and station keeping analyses. Assuming that the spacecraft completes its mission with no propellant, a final spacecraft mass is calculated using the user specified spacecraft dry mass and an arbitrarily chosen tankage mass. This initial spacecraft is flown "backwards" through the mission, adding the propellant required for each orbit raising/lowering defensive maneuver and each on-station leg. After iterating "backwards" for the time required for the mission duration, initial estimates for the required propellant and tankage mass are obtained. This initial spacecraft mass is then subject to the same iterative scheme used by the orbit raising mission analysis.

3.7 Reference Search Utility

SPACEDRIVE allows the user access to an electric propulsion reference library through either predefined keywords or an author's name. Once the user has selected keywords (up to five) or given an author's name, the entire reference database is searched in one pass, examining keyword or author name data fields for matches. Each reference contained in the reference database has five keyword data fields and three author name data fields and when the data in the appropriate fields agrees with the user's input, the reference is displayed. There are approximately 1100 references recorded in this electric propulsion reference database.

Keywords are grouped according to topics. If two keywords are selected from the same topic, they are concatenated with the logical "or" operator. In this case, references containing either of the specified keywords are selected. When two keywords are selected from different topics, they are concatenated with the logical "and" operator. In this case, only references containing both keywords are selected. By selecting keywords from topic lists, the user can construct very specific searches of the reference data base.

3.8 Overviews Utility

SPACEDRIVE contains descriptive overviews of many electric propulsion engine concepts that have been developed, or are presently under development. Each overview describes the basic operating principles of the engine concept, typical performance levels, development history, its scaleability to higher power levels, and any flight tests. It is highly recommended that the user read the overviews provided in this utility to obtain a better understanding of electric propulsion engines for more effective use of the other SPACEDRIVE utilities.

4.0 SPACEDRIVE USER'S GUIDE

Potential users of the SPACEDRIVE software include personnel that wish to acquire a more complete understanding of the capabilities of electric propulsion engines, as well as individuals who require immediate projections on the electric propulsion system design required for a specific application or the mission capability of a specific engine design. Individuals who wish to canvas existing reports, articles or papers for general knowledge or specific performance data will find the library reference search useful. Finally, the overviews included in SPACEDRIVE can be used for educational purposes, or as a quick reference to general engine capabilities of the many electric propulsion concepts presented.

4.1 SPACEDRIVE Installation

Installation of the SPACEDRIVE software requires a 640K IBM PC, XT, AT, or 100% compatible computer with at least one floppy disk drive and a hard disk with 1.0MB of free space. Also, the computer must use either MS DOS 3.2 (or higher) or PC DOS 3.2 (or higher) and must have files = 20 in the Config.sys file. The computer system must also use an EGA card and color monitor. To install SPACEDRIVE, the user should create a subdirectory (the name is arbitrary) on his hard disk and then copy all the files from all of the SPACEDRIVE diskettes into this subdirectory. After completing this installation, the user enters his subdirectory containing the SPACEDRIVE software and types "SPACEDRIVE" at the DOS prompt. SPACEDRIVE then begins execution. The color monitor may be adjusted to display the SPACEDRIVE menus in a blue-green-brown palette, or to the user's taste.

4.2 SPACEDRIVE Operation

User interactive menus drive the SPACEDRIVE software. In the top level menu, the user must select one of the four utilities: a) systems analysis, b) mission analysis, c) reference library search, or d) overview display. As with all the SPACEDRIVE menus, "HELPS" are displayed, instructing the user on how to make his selection. In addition, the menus are color coded to assist the user in understanding and remembering the organization of the information on the screen. After the user responds to the selection query, the system, mission, reference, or overview menu appears, as requested by the user. Each of these menus requires some inputs from the user. If the user should make a selection or enter data SPACEDRIVE cannot process, the unacceptable input is rejected, the user is informed of the reason for rejection, and he is required to modify his entry before the program can continue. Once all inputs are entered, the user is requested to review his data and is given the opportunity to change any entries.

When data entry is completed, the user begins the analysis, search or display with a keystroke entry. The user must wait until the utility calculations are completed. Results are automatically displayed on the screen. This calcuation waiting period depends on the capabilities of the host computer. In all cases, availability of a math coprocessor will significantly reduce SPACEDRIVE execution time. In general, for an IBM XT or compatible computer operating at 8MHz with a math coprocessor, most system analysis results are displayed in a few seconds while a few minutes of waiting time is typical before most mission analysis and reference search results are displayed. However, for certain mission analysis scenarios the waiting time can be of the order of half an hour. (In the event of a non-converging calculation, a counter in the mission analysis model will stop program execution after approximately two hours). At this point, the user has the options of continuing or terminating his SPACEDRIVE session.

4.2.1 Helps

As stated above, each SPACEDRIVE menu is accompanied with screen-displayed "HELPS". These helps state the restrictions on SPACEDRIVE acceptable data. For the entry of numerical data, these "HELPS" are of particular importance because they define units of measure and the bounds for acceptable data. It is very important for the user of SPACEDRIVE to read these "HELPS" while he enters data.

4.2.2 Printer Instructions

When the user chooses to print his results, he is instructed to check his printer and paper. At this point, it is very important that the user ensure that his printer is on-line and that sufficient paper is available. If these checks are not completed correctly, SPACEDRIVE may stop execution, or behave in some other unexpected manner.

4.2.3 Terminating a SPACEDRIVE Session

At any time that the user is requested to make a selection or enter data, he may terminate his SPACEDRIVE session by pressing the "ESC" key. Pressing the "ESC" key will not terminate the session if a system analysis, mission analysis, reference library search or overview display is in progress. Because the analyses and searches can be time consuming (several minutes), the user is urged to verify his data before beginning an analysis or search. It will be necessary to reboot the computer to reenter SPACEDRIVE if the user wants to terminate the progress of a lengthy calculation or search.

4.2.4 Disclaimer and Software Protection

The Electric Propulsion Laboratory, Inc. provides SPACEDRIVE without warranty of any kind. It is a virtual certainty that the software contains some bugs, despite thorough testing. In addition, this software is protected by the DoD FAR subparagraph C1(ii) Rights and Technical Data and Computer Software Clause at 52.227-7013. The user should not duplicate or disclose this software without the explicit permission of the Electric Propulsion Laboratory, Inc. In the event that an accompanying diskette is corrupted, please contact the Electric Propulsion Laboratory, Inc.

4.3 SPACEDRIVE Code

SPACEDRIVE uses two programming languages, dBaseIII+, compiled with Clipper, and Microsoft FORTRAN. The dBaseIII+, Clipper compiled code is used primarily to present the user interactive menus, to display the results, to perform the library searches and to record the reference library and overview data. Microsoft FORTRAN is used to program the routines that perform the system and mission analyses. Data that the user enters in the interactive menus is passed to the FORTRAN routines via ASCII data files. Similarly, analysis results are passed from the FORTRAN routines to the dBaseIII+ display routines via ASCII data files. It is this transfer of data that links the operation of the different sections of the SPACEDRIVE code.

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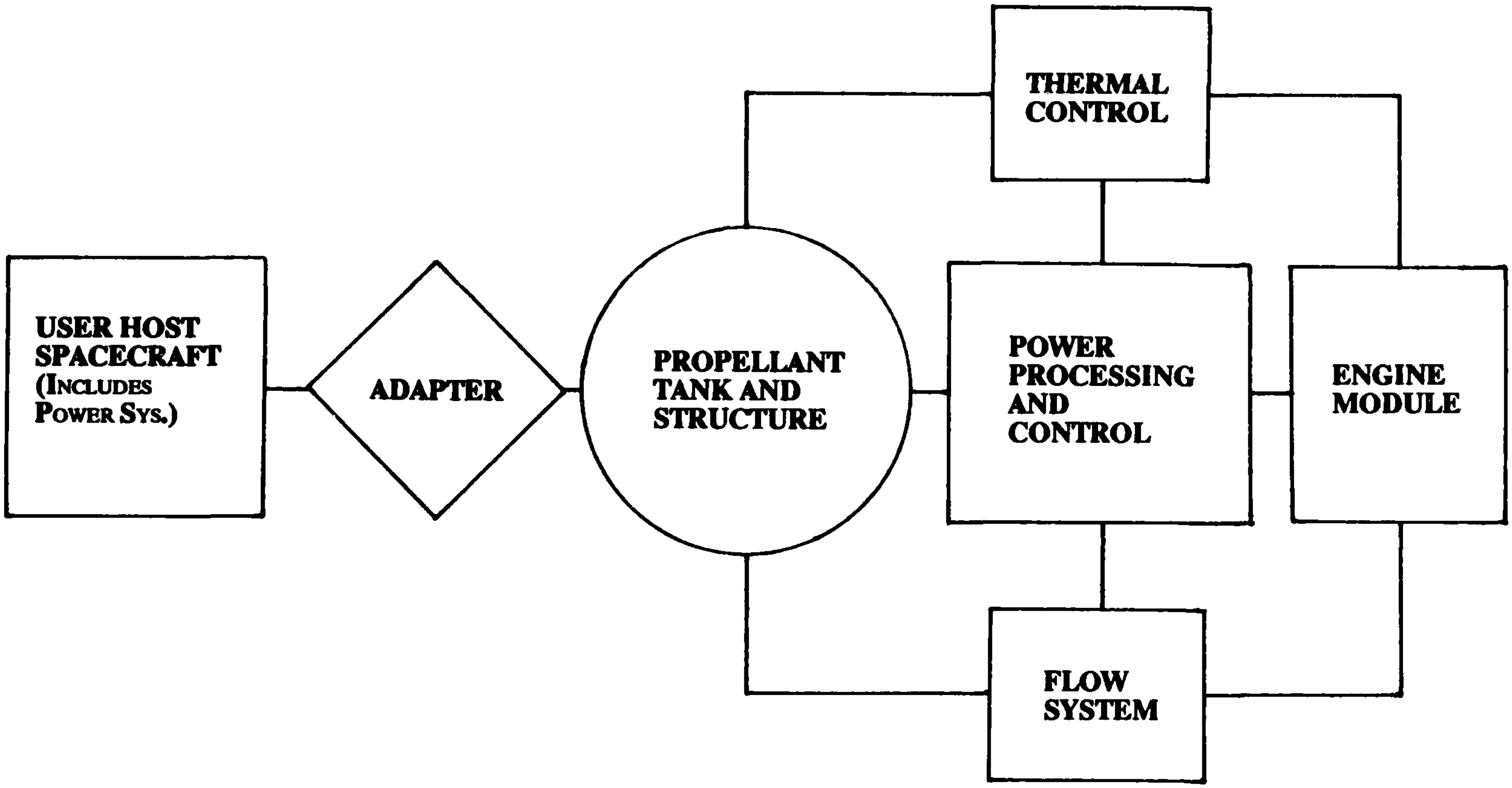

Figure 1. Major Electric Propulsion System Elements.

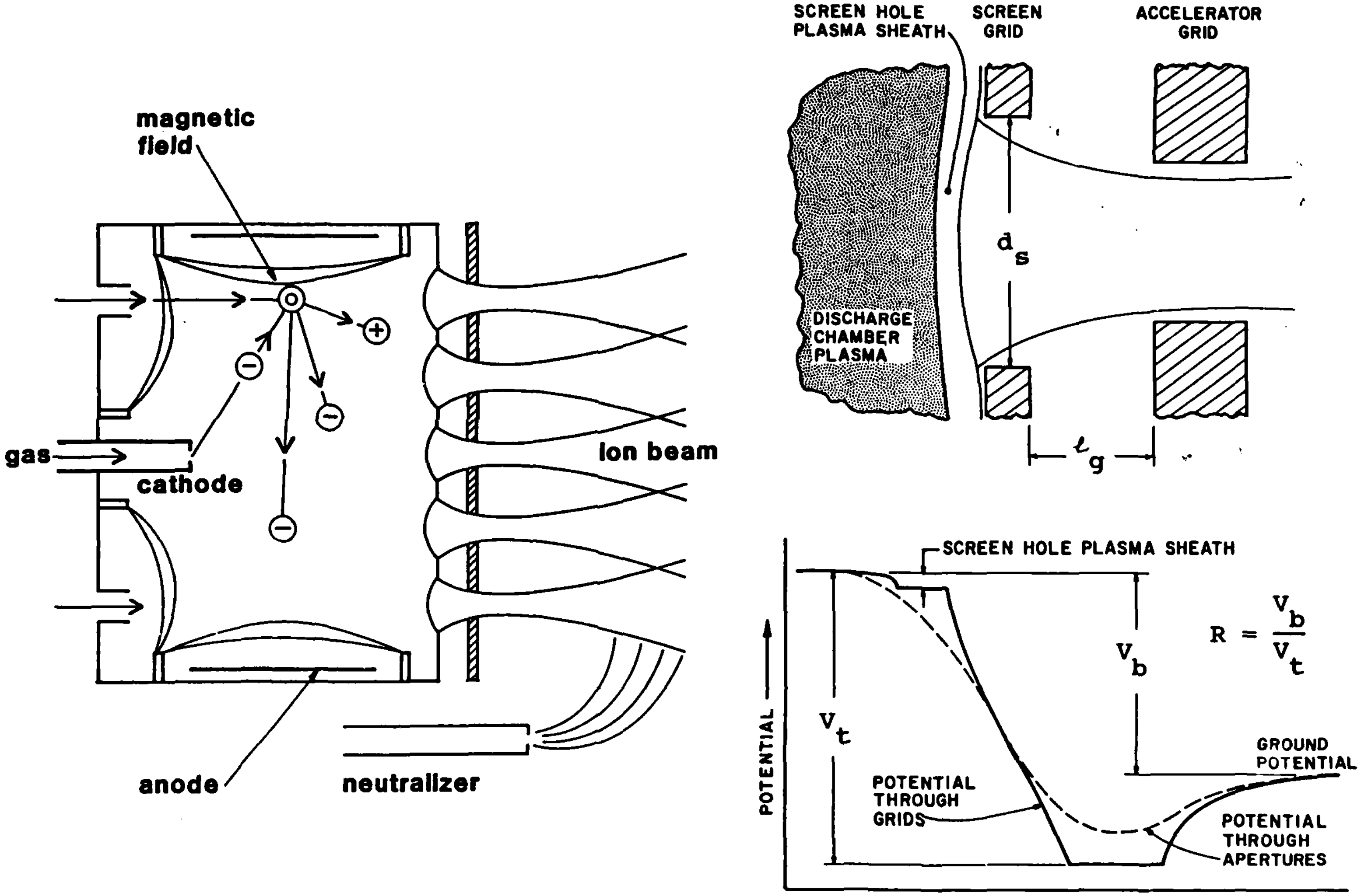

Figure 2. Electron-bombardment Ion Engine.

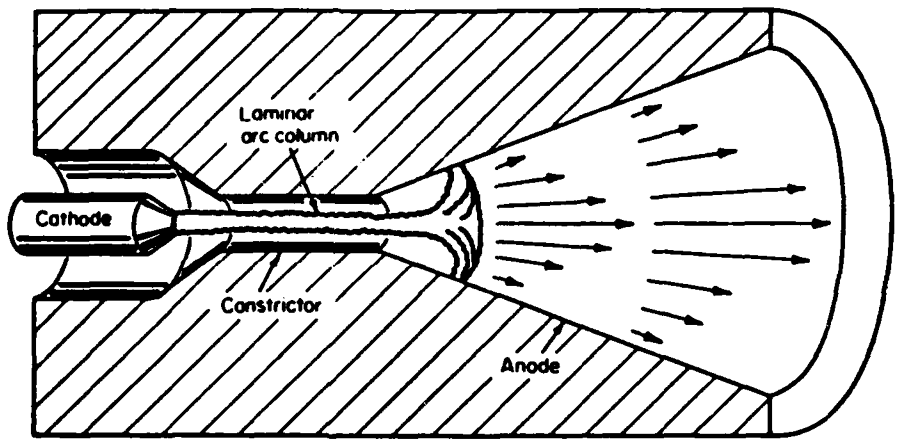

Figure 3. D.C. Arcjet Engine.